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ALTITUDE EVALUATION OF THE INTELSAT II COMMUNICATIONS SPACECRAFT APOGEE MOTOR IN THE SPIN MODE AT AN ENVIRONMENTAL TEMPERATURE OF 55°F

A. A. Cimino and C. W. Stevenson
ARO, Inc.

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ROCKET TEST FACILITY
ARNOLD ENGINEERING DEVELOPMENT CENTER
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ARNOLD AIR FORCE STATION, TENNESSEE

NOTICES

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FOREWORD

The test program reported herein was sponsored by the National Aeronautics and Space Administration (NASA) for the Communications Satellite Corporation (COMSAT) under Program Area Number 921E, Project Number 9240. Test direction was accomplished by COMSAT with technical assistance from Hughes Aircraft Company (HAC) and Aerojet-General Corporation (AGC).

The results of the test were obtained by ARO, Inc. (a subsidiary of Sverdrup & Parcel and Associates, Inc.), contract operator of the Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC), Arnold Air Force Station, Tennessee, under contract AF 40(600)-1200. The test was conducted in Propulsion Engine Test Cell (T-3) of the Rocket Test Facility (RTF) from December 16 to 19, 1966, under ARO Project Number RC1719, and the manuscript was submitted for publication on February 10, 1967.

This technical report has been reviewed and is approved.

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ABSTRACT

An Aerojet-General Corporation SVM-1 solid-propellant apogee rocket motor installed in an INTELSAT II communications spacecraft (S/N T-1) was fired, while spinning at 120 rpm, at a pressure altitude of 108,000 ft. The primary objective of the program was verification of vacuum performance of the apogee motor after a 75-hr exposure of the motor to a 55 ± 5°F temperature and a near-vacuum conditioning environment. Secondary objectives were to evaluate the control and performance of a strip heater attached to the nozzle flange, to check the structural integrity of a thermal shield on the motor aft dome, and to determine the existence of any significant dynamic coupling between the motor and the spacecraft. Motor performance is presented and compared with data from earlier firings of the same type of motor. Temperature-time histories at selected locations on the motor and spacecraft are also presented. Performance of the nozzle flange strip heaters, the structural integrity of the thermal shield, and vibration data are also discussed.

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NOMENCLATURE

A_e Nozzle exit area, in.^{2}

A_t Nozzle throat area, in.^{2}

C_f Average thrust coefficient, based on total burn time,

$$\frac{I_{vac}}{A_{t_{avg}} \int_{t_t} P_{ch}\, dt}$$

F Motor axial thrust, lb_f

I_{sp}	Specific impulse based on the manufacturer's stated propellant weight and total burn time, lb_f-sec/lb_m
I_{vac}	Vacuum impulse based on total burn time, lb_f-sec, $\int_{t_t} F dt + A_{e_{avg}} \int_{t_t} P_{cell} dt$
P_{cell}	Test cell pressure, psia
P_{ch}	Motor chamber pressure, psia
t_0	Zero time, time at which firing voltage is applied to the igniter circuit, sec
t_a	Action time, time interval between 10 percent of maximum chamber pressure during ignition and 10 percent of maximum chamber pressure during tailoff, sec
t_ℓ	Ignition lag time, time interval from zero time to time of increase in chamber pressure, sec
t_t	Total burn time, interval from time of increase in chamber pressure during ignition until chamber pressure has decreased to cell pressure during tailoff, sec
W	Manufacturer's stated propellant weight, lb_m

SECTION I
INTRODUCTION

The Aerojet-General Corporation (AGC) SVM-1 solid-propellant rocket motor is utilized as the apogee motor in the Communications Satellite Corporation (COMSAT) INTELSAT II communications satellite (Fig. 1, Appendix I). INTELSAT II is a spin-stabilized, active, communications satellite for use in synchronous equatorial orbits. The spacecraft program objective is to increase the commercial communications capacity across the Atlantic and Pacific Oceans. The AGC SVM-1 motor will provide the final velocity increment required at the apogee of the spacecraft elliptical transfer orbit to place the satellite into a synchronous equatorial orbit (Ref. 1).

As part of the apogee motor development and qualification test program, 16 motors were successfully fired during earlier tests at the Aerojet facility (Ref. 2). Fourteen of these motors were fired in a nonspin mode. The remaining two were fired while spinning about the motor axial centerline at 120 rpm.

After the first INTELSAT II spacecraft launch, the apogee motor, scheduled to place the spacecraft in a synchronous orbit, thrusted for approximately 5 sec (normal burn time is 17 sec) and then terminated. Analysis of telemetry data indicated a failure of the nozzle attachment flange which allowed the nozzle assembly to separate and depressurize the motor chamber (Ref. 3). Subsequent investigations and calculations indicated that, during the transfer orbit, the aft portion of the apogee motor case was subjected to a much colder (-28°F) environment than that for which the motor had been qualified. A motor was conditioned to the temperatures which had been encountered and fired in an altitude chamber at the AGC facility. Shortly after ignition, the nozzle assembly separated from the motor case. Subsequent examination indicated a low-temperature-induced failure of the motor insulator, permitting a flame path to the fiber glass case around the periphery of the nozzle flange.

The proposed solution to the problem was to install a radiation shield over the motor aft dome and to provide a strip heater around the nozzle flange in order to maintain temperatures in these areas above the minimum qualification level (Ref. 3).

This report presents the results of a near-vacuum test firing at the AEDC of the apogee rocket motor mounted in a flightweight INTELSAT II spacecraft. This motor incorporated both the radiation heat shield and a 10-w strip heater on the nozzle flange. The motor was fired with the spacecraft spinning about the longitudinal axis at 120 rpm after the assembly was exposed to near vacuum conditions and to a controlled temperature environment of 55 ± 5°F for 75 hr.

The primary objective of the test program was verification of the vacuum performance of the apogee motor after exposure to the expected transfer orbit conditions. Secondary objectives were to evaluate the control and performance of the nozzle flange strip heater, to check the structural integrity of the thermal shield, and to determine the existence of any significant dynamic coupling between the motor and the spacecraft. Altitude ignition characteristics and ballistic performance of the motor are presented. Motor and spacecraft temperatures and structural integrity are discussed. Performance of the nozzle flange strip heaters, the structural integrity of the thermal shield, and vibration data are also discussed.

SECTION II
APPARATUS

2.1 TEST ARTICLE

The AGC SVM-1 solid-propellant rocket motor (Fig. 2) used in this test was a flight motor which had been scheduled for launch. Nominal motor characteristics are:

Length, in.	32
Diameter, in.	18
Loaded Weight, lb_m	193
Propellant Weight, lb_m	163
Maximum Thrust, lb_f	3300
Maximum Chamber Pressure, psia	400
Burn Time, sec	17
Throat Area, in.^{2}	4.23
Nozzle Area Ratio, A/A^*	33.25

Motor weight and physical dimension data are presented in Table I (Appendix II).

The cylindrical motor case is constructed of glass-filament roving and epoxy resin, with aluminum bosses at each end which provide attachment points for the igniter and nozzle. An aluminum flange provides for attachment to the satellite. The case is thermally protected and sealed against leaks by a molded silica-filled butadiene acrylonitrile rubber (Gen-Gard V-45) insulator.

The contoured nozzle assembly contains a carbon-phenolic entrance section, a silver-infiltrated tungsten throat insert, and an ablative exit cone constructed of silica cloth-phenolic tape and a glass-cloth/epoxy-resin overwrap. The nozzle assembly has a nominal area ratio of 33.25:1 and a 17-deg half-angle at the exit plane. A plug in the nozzle prevented foreign matter from entering the motor case.

The motor contains an ANB-3066 solid-propellant (ICC Class B), composed of ammonium perchlorate (73-percent), aluminum (15-percent), and a carboxy-terminated polybutadiene binder (12-percent). The grain design has a neutral burning conocyl configuration (a conical forward section and cylindrical center and aft sections). The ratio of specific heats (assuming frozen equilibrium) of the propellant exhaust gases is 1.18.

The igniter (Fig. 3) consists of a primary-charge assembly contained in a pressure vessel, a main-charge assembly positioned around the pressure vessel, and a squib assembly which is inserted into the pressure vessel. The pressure vessel is made of a glass-filament and epoxy-resin laminate. The primary charge is constructed of a copper-screen inner-flame tube, centered in the cellulose acetate butyrate (CAB) primary housing. Boron potassium nitrate (BPN) pellets are contained in the flame-tube-housing annulus. The main-charge assembly is constructed of concentric CAB cylinders to produce an annulus which contains 130 gm of BPN pellets. The squib assembly consists of an electrical connector, two Flare-Northern F/ND-209 squibs (1 w/1 amp/5 min, no-fire), associated wiring, and an RF shield. The two squibs were wired in parallel with a single power supply. The igniter used in this test, unlike the flight igniters, had a chamber pressure tap.

The motor was mounted in a flightweight engineering test spacecraft assembly with inactive electronic components. The microwave antenna was removed to simplify the test installation. A 10-w strip heater was installed around the nozzle flange and a Hughes-designed aluminum foil radiation shield was installed over the motor aft dome to match the flight configuration. The motor was fired while mounted in the spacecraft with the entire assembly spinning at 120 rpm.

2.2 INSTALLATION

The motor spacecraft assembly was cantilever mounted from the aft bearing of a spin fixture assembly in Propulsion Engine Test Cell (T-3). The spin assembly was mounted on a thrust cradle, which was supported from the cradle support stand by three vertical and two horizontal double-flexure columns (Fig. 4). The spin fixture assembly consisted of a 10-hp squirrel-cage-type drive motor, a forward thrust bearing assembly, a drive shaft and thrust pylon, and an aft bearing assembly. The spin fixture rotates counterclockwise, looking upstream. Electrical leads to and from the igniter, pressure transducers, thermocouples, and accelerometers on the rotating motor were provided through two 52-channel slip-ring assemblies mounted on the drive shaft. A counter-balance weight was mounted on the rotating shaft between the bearing assemblies to reduce the thrust column bending moment at the downstream slip-ring assembly. Axial thrust was transmitted through the drive shaft-thrust bearing assembly to two load cells mounted just forward of the thrust bearing.

Pre-ignition pressure altitude was maintained in the test cell by a steam ejector operating in series with the RTF exhaust gas compressors. During motor firing, the motor exhaust gases were used as the driving gas for the 30.0-in.-diam, ejector-diffuser system to maintain test cell pressure at an acceptable level.

2.3 INSTRUMENTATION

Instrumentation was provided to measure axial thrust; test cell and motor chamber pressures; nozzle, motor case, and spacecraft temperatures; motor and spacecraft vibration levels; and motor-spacecraft rotational speed. Table II presents instrumentation ranges, recording methods, and system accuracies for all reported parameters.

The axial thrust measuring system consisted of two double-bridge, strain-gage-type load cells mounted in the axial double-flexure column forward of the thrust bearing on the rocket motor centerline.

Unbonded strain-gage-type transducers were used to measure test cell pressure. Bonded strain-gage-type transducers were used to measure motor chamber pressure. Copper-constantan (CC) thermocouples were bonded to the motor case, nozzle, and spacecraft to measure temperatures before, during, and after motor firing. Piezoelectric-type accelerometers were used to measure the dynamic effects of the apogee motor on the spacecraft. Thermocouple and accelerometer locations are presented in Table III. Rotational speed of the motor and spin rig assembly was determined from the output of a magnetic pickup.

The output signal of each measuring device was recorded on independent instrumentation channels. Primary data were obtained from four axial-thrust channels, four test cell pressure channels, and two motor chamber pressure channels. These data were recorded as follows: Each instrument output signal was indicated in totalized digital form on a visual readout of a millivolt-to-frequency converter. A magnetic tape system, recording in frequency form, stored the signal from the converter for reduction at a later time by an electronic digital computer. The computer provided a tabulation of average absolute values for each 0.10-sec time increment and total integrals over the cumulative time increments.

The output signal from the magnetic rotational speed pickup was recorded in the following manner: A frequency-to-analog converter was triggered by the pulse output from the magnetic pickup and in turn supplied a square wave of constant amplitude to the electronic counter, magnetic tape, and oscillograph recorders. The scan sequence of the electronic counter was adjusted so that it displayed directly the motor spin rate in revolutions per minute.

The millivolt outputs of the thermocouples were recorded on a multipoint, null-balance, potentiometer-type strip chart at a sampling rate of 2 samples/min/thermocouple during the altitude coast period and were recorded on magnetic tape from an analog-to-digital converter at a sampling rate for each thermocouple of 50 samples/sec during the motor firing. The millivolt outputs of the accelerometers were conditioned through individual charge amplifiers prior to transmission through slip rings. The amplified outputs were recorded on FM analog magnetic tape and played back to a photographically recording, galvanometer-type oscillograph at a later time.

A recording oscillograph was used to provide an independent backup of all operating instrumentation channels except the temperature and accelerometer systems. Selected channels of thrust and pressures were recorded on null-balance, potentiometer-type strip charts for analysis immediately after a motor firing. Visual observation of the firing was provided by a closed-circuit television monitor. High-speed, motion-picture cameras provided a permanent visual record of the firings.

2.4 CALIBRATION

The thrust calibration weights, thrust load cells, pressure transducers, and accelerometers were laboratory calibrated prior to usage in this test. After installation of the measuring devices in the test cell, thrust load cells were again calibrated at sea-level, nonspin ambient conditions and at simulated altitude while spinning at 120 rpm.

The pressure recording systems were calibrated by an electrical, four-step calibration, using resistances in the transducer circuits to simulate selected pressure levels. The axial thrust instrumentation systems were calibrated by applying to the thrust cradle known forces, which were produced by deadweights acting through a bell crank. The calibrator is hydraulically actuated and remotely operated from the control room. Thermocouple and accelerometer recording instruments were calibrated by using known millivolt levels to simulate thermocouple and accelerometer outputs.

After each motor firing, with the test cell still at simulated altitude pressure, the recording systems were recalibrated to determine any shift.

SECTION III
PROCEDURE

The AGC SVM-1 solid-propellant rocket motor arrived at AEDC on December 10, 1966. The motor was visually inspected for possible shipping damage and radiographically inspected for grain cracks, voids, or separations and found to meet criteria provided by the manufacturer. During storage in an area temperature conditioned at 75 ± 5°F, the motor and spacecraft were checked to ensure correct fit of mating hardware, and the electrical resistance of the igniter was measured. The nozzle throat and exit diameters were measured, the entire motor assembly was weighed and photographed, and the thermocouples and accelerometers were installed. The chamber pressure instrumentation and squib assembly were installed, and a pressure leak test was conducted on the motor. Motor and spacecraft surfaces were mated and aligned. The nozzle plug was removed to accomplish the pre-fire inspection and leak test, then replaced, and punctured for the firing.

After installation of the motor-spacecraft assembly in the test cell, the spacecraft centerline was axially aligned with the spin axis by rotating the motor-spacecraft assembly and measuring the deflection of spacecraft thrust tube flange surfaces with a dial indicator. Instrumentation connections were made, and a balance check was performed by rotating the assembly at a speed of 120 rpm. Trichloroethane was loaded into the H_2O_2 tanks to simulate the flight condition. An annular shield was installed around the nozzle exit cone near the diffuser entrance to protect the motor against possible hot gas recirculation back into the test cell during thrust tailoff. The telemetry whip antennas were installed and a continuity check of all electrical systems was performed.

Pre-fire, sea-level calibrations were completed, the test cell pressure was reduced to the desired pressure altitude condition, and spinning of the unit was started. After spinning had stabilized at 120 rpm, a complete set of altitude calibrations was taken. With the test cell pressure still at the desired pressure altitude condition, motor-spacecraft assembly spinning was stopped and a 75-hr pressure altitude soak period was begun (Fig. 5).

Test cell temperature was varied to maintain the temperature of the control thermocouples on the spacecraft (T/C Nos. 18 and 31) at 55 ± 5°F (Fig. 6). The electrical power input to the nozzle flange strip heater was controlled to maintain the temperature of the nozzle flange (T/C Nos. 3 and 9) at 70 ± 5°F (Figs. 7 and 8). As the pressure altitude soak period ended, the motor-spacecraft assembly was again spun at 120 rpm, and a complete set of altitude calibrations was taken.

The final operation prior to firing the motor was to adjust the firing circuit to provide the desired 4.5-amp current to each of two igniter squibs. The entire instrumentation measuring-recording complex was activated, and the motor was fired while spinning (under power) at 120 rpm.

Spinning of the motor was continued for approximately 35 min after burnout, during which time post-fire calibrations were accomplished. The unit was decelerated slowly until rotation had stopped, and another set of calibrations was taken. The test cell pressure was then returned to ambient conditions, and the motor and spacecraft were inspected, photographed, and removed to the storage area. Post-fire inspections at the storage area consisted of measuring the throat and exit diameters of the nozzle, weighing the motor, and photographically recording the post-fire condition of the motor.

SECTION IV
RESULTS AND DISCUSSION

A test program was conducted in Propulsion Engine Test Cell (T-3) utilizing a flightweight INTELSAT II spacecraft (S/N T-1) containing an Aerojet-General Corporation SVM-1 solid-propellant rocket motor. The motor was fired at a pressure altitude of 108,000 ft with the spacecraft spinning about the longitudinal axis at 120 rpm. The spacecraft and motor were exposed to a near vacuum and a controlled temperature environment of 55 ± 5°F for 75 hr prior to firing the motor. The primary objective of the test program was verification of the apogee motor vacuum performance after exposure of the motor to a 55 ± 5°F temperature and a near-vacuum conditioning environment for 75 hr. Secondary objectives were to evaluate the control and performance of the nozzle flange strip heater, to check the structural integrity of the thermal shield, and to determine the existence of any significant dynamic coupling between the motor and the spacecraft. The resulting data are presented in both tabular and graphical form.

Altitude ignition characteristics and ballistic performance of the motor are presented. Motor and spacecraft temperatures and structural integrity are discussed. Performance of the nozzle flange strip heaters, the structural integrity of the thermal shield, and vibration data are also discussed.

Motor performance data based on total burn time (t_t) are summarized in Table IV. The average measured total impulse was corrected to vacuum conditions by adding to it the product of the cell pressure integral and the average of the pre- and post-fire nozzle exit area. The average vacuum correction was approximately 0.61 percent of the average measured total impulse. Specific impulse values are presented using both the manufacturer's stated propellant weight and the motor expended mass determined from AEDC pre- and post-fire motor weight. When more than one instrumentation channel was used to obtain values of a single parameter, the average value is discussed and used to calculate the data presented.

4.1 ALTITUDE IGNITION CHARACTERISTICS

The motor was successfully ignited at a pressure altitude of 104,000 ft. An analog trace of the ignition event is shown in Fig. 9. Ignition lag time, defined as the time interval from the time at which firing voltage is applied to the igniter circuit to the first indication of a rise in chamber pressure, was 0.015 sec.

4.2 BALLISTIC PERFORMANCE

Variations of thrust, chamber pressure, and cell pressure during the motor firing are shown in Fig. 10. Action time (t_a) was 16.6 sec. Total burn time (t_t) was 17.7 sec. Vacuum corrected total impulse was 46,931 lb_f-sec. Vacuum specific impulse, based on the manufacturer's stated propellant weight, was 287.87 lb_f-sec/lb_m. Vacuum specific impulse, based on the expended mass, was 283.56 lb_f-sec/lb_m. The average vacuum thrust coefficient, based on the average of pre- and post-fire throat areas, was 1.839.

A comparison of the vacuum thrust from the HAC performance specification (Ref. 4), from previous firings at AGC (Ref. 5), and from the test reported herein is presented in Fig. 11. The conditions affecting the measured performance levels are as follows:

	Spinning at 120 rpm	Altitude Firing	Conditioning Temperature, °F
HAC Specification	x	x	55
AGC (Motor S/N QA-1)	Nonspin	x	30
AGC (Motor S/N QA-4)	Nonspin	x	70
AEDC (Motor S/N A-22)	x	x	55 to 70

4.3 MOTOR CASE AND SPACECRAFT TEMPERATURES

Thermocouples were bonded to the motor case and spacecraft (Table III) to measure temperatures during the 75-hr altitude coast period, the firing, and the 35-min period following the firing. Temperature-time histories from selected thermocouples on the motor and spacecraft are presented in Fig. 12.

The maximum temperatures recorded were:

Location	Temperature, °F	Time after Ignition, min
Nozzle exit cone (T-10)	525	2
Nozzle attachment flange (T-9)	373	13
Motor case wall, aft closure (T-6)	245	20
Motor case wall, forward of attachment flange (T-12)	296	2
Spacecraft thrust tube (T-25)	135	11

Nozzle flange temperature was maintained at 70 ± 5°F during the 75-hr pre-fire period by varying the power input to the strip heater (Fig. 7). The average electrical power input required was approximately 6.5 w (Fig. 8).

4.4 STRUCTURAL INTEGRITY

Post-fire photographs of the motor case and nozzle are presented in Fig. 13. Delamination of the glass cloth overwrap was evident along the external surface of the nozzle extension.

Pre- and post-fire inspection data revealed a decrease in throat area of approximately 0.77 percent during the firing. The decrease in nozzle exit area was approximately 0.32 percent during the firing (exhaust product deposition was not removed prior to throat and exit diameter measurements). There was no evidence of motor case, radiation shield, or spacecraft deterioration. Structural integrity of all test components is considered adequate.

4.5 VIBRATION

Vibrational data from locations on the nozzle and motor body during ignition are presented in Fig. 14. Analysis of the accelerometer data indicated that, after motor ignition with its associated high frequency transients, motor operation was extremely smooth. Peak values of 17.7 g's are identified at a frequency of 120 cps on the nozzle and 1.4 g's at a frequency of 130 cps on the motor body.

SECTION V
SUMMARY OF RESULTS

One Aerojet-General Corporation SVM-1 solid-propellant rocket motor was fired at a pressure altitude of 108,000 ft while mounted in a flightweight INTELSAT II spacecraft which was spinning at 120 rpm about the spacecraft axis. The spacecraft and motor were conditioned to a temperature of 55 ± 5°F and exposed to near vacuum condition for 75 hr prior to firing the motor. The results are summarized as follows:

1. The time interval from the time at which firing voltage is applied to the igniter circuit to the first indication of a rise in chamber pressure was 0.015 sec.

2. The time interval from the time of increase in chamber pressure during ignition until chamber pressure has decreased to cell pressure during tailoff (t_t) was 17.7 sec.

3. Vacuum total impulse based on t_t was 46,931 lb_f-sec. Vacuum specific impulse, based on the manufacturer's stated propellant weight and t_t, was 287.87 lb_f-sec/lb_m.

4. The average vacuum thrust coefficient, based on t_t and the average of pre- and post-fire throat areas, was 1.839.

5. An average power input to the nozzle flange strip heater of 6.5 w was required to maintain a temperature of 70 ± 5°F on the nozzle flange during the 75-hr pre-fire altitude coast period.

6. The maximum nozzle exit cone temperature was approximately 525°F and occurred approximately 2.0 min after ignition. The maximum nozzle attachment flange temperature was 373°F, occurring 13 min after ignition.

7. The nozzle throat area decreased 0.77 percent, and the nozzle exit area decreased 0.32 percent during firing.

8. Peak values of 17.7 g's are identified at a frequency of 120 cps on the nozzle and 1.4 g's at a frequency of 130 cps on the motor body.

9. Post-fire inspection verified the structural integrity of the thermal radiation shield.

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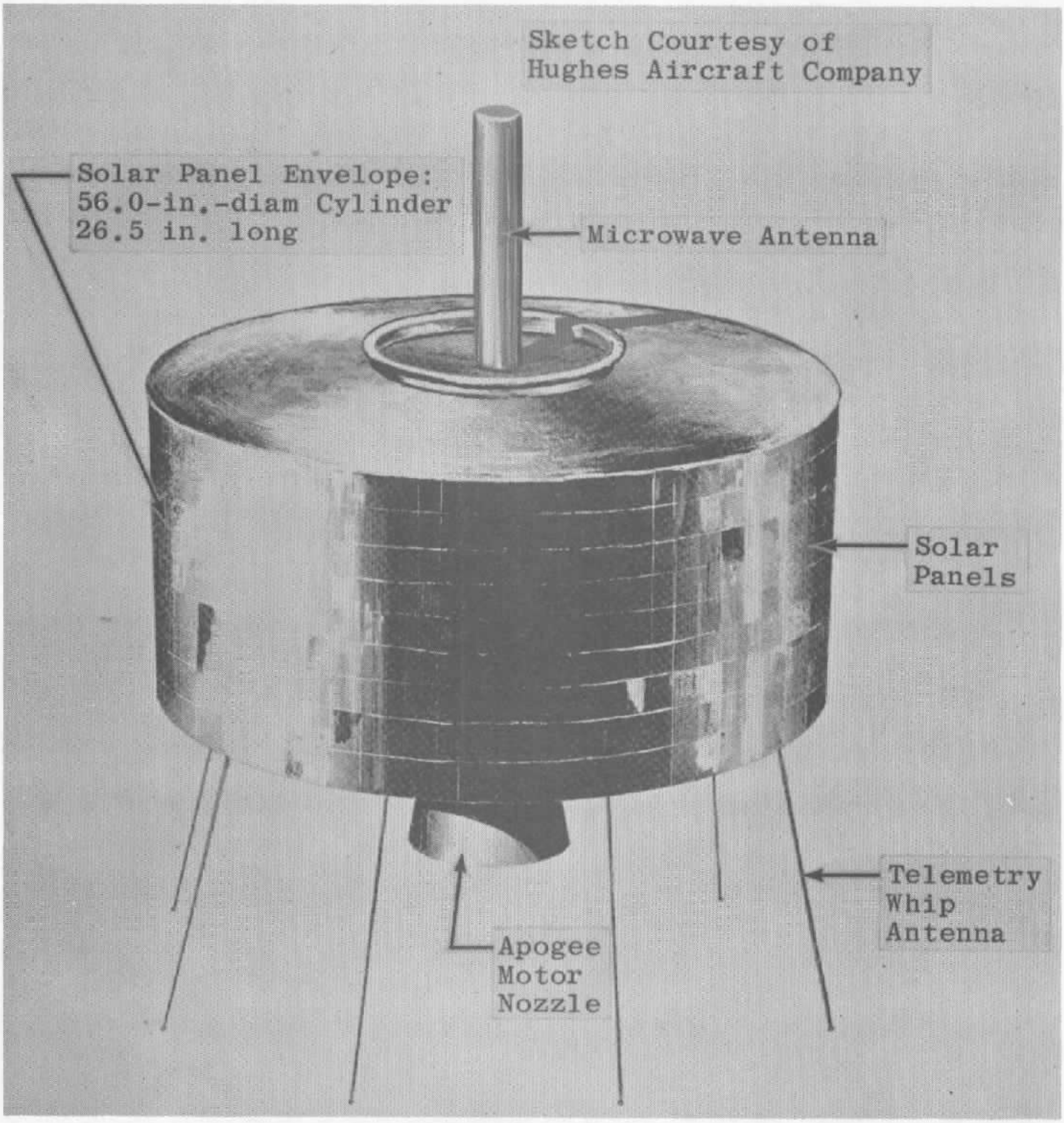
APPENDIXES

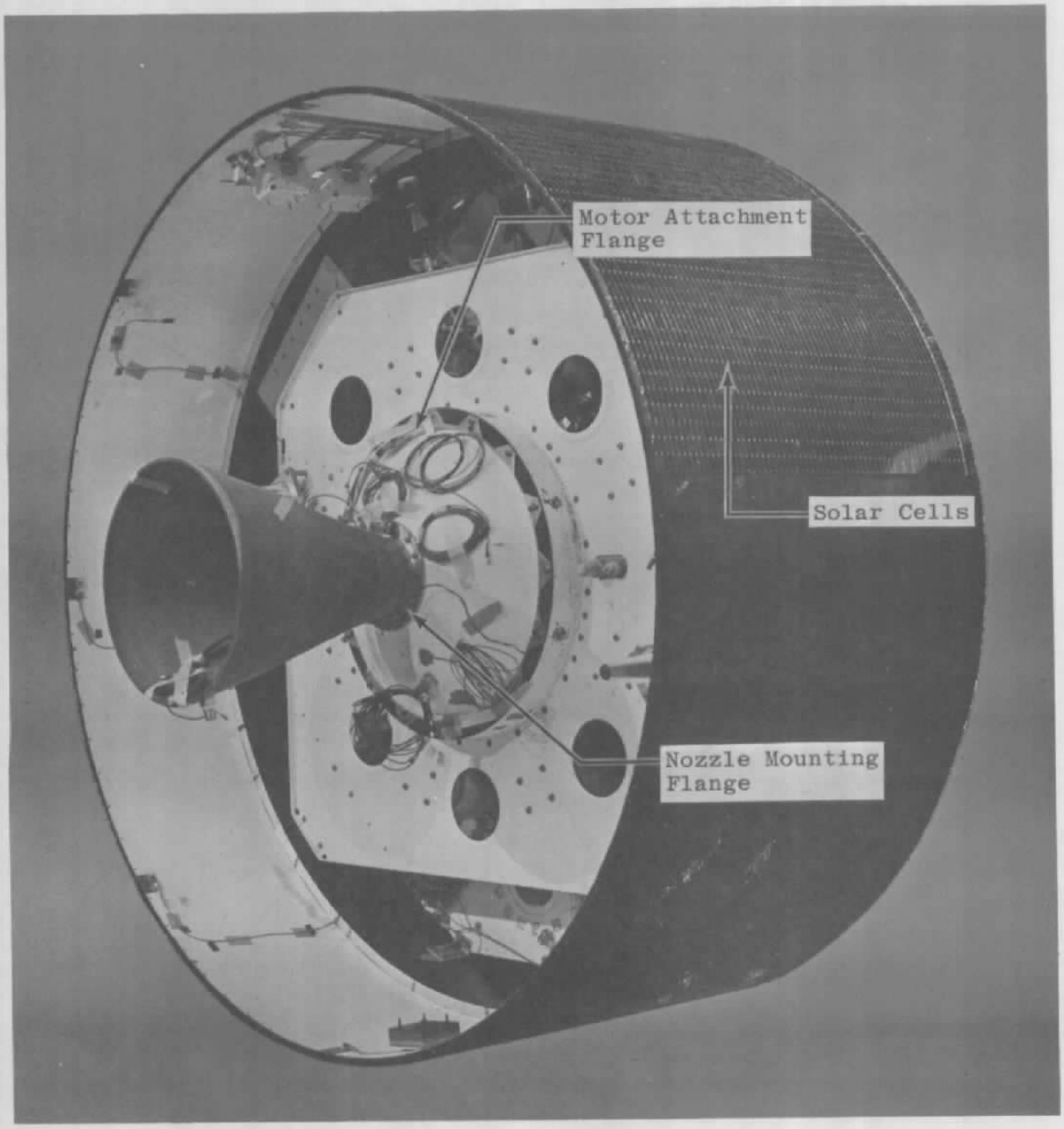
I. ILLUSTRATIONS

II. TABLES



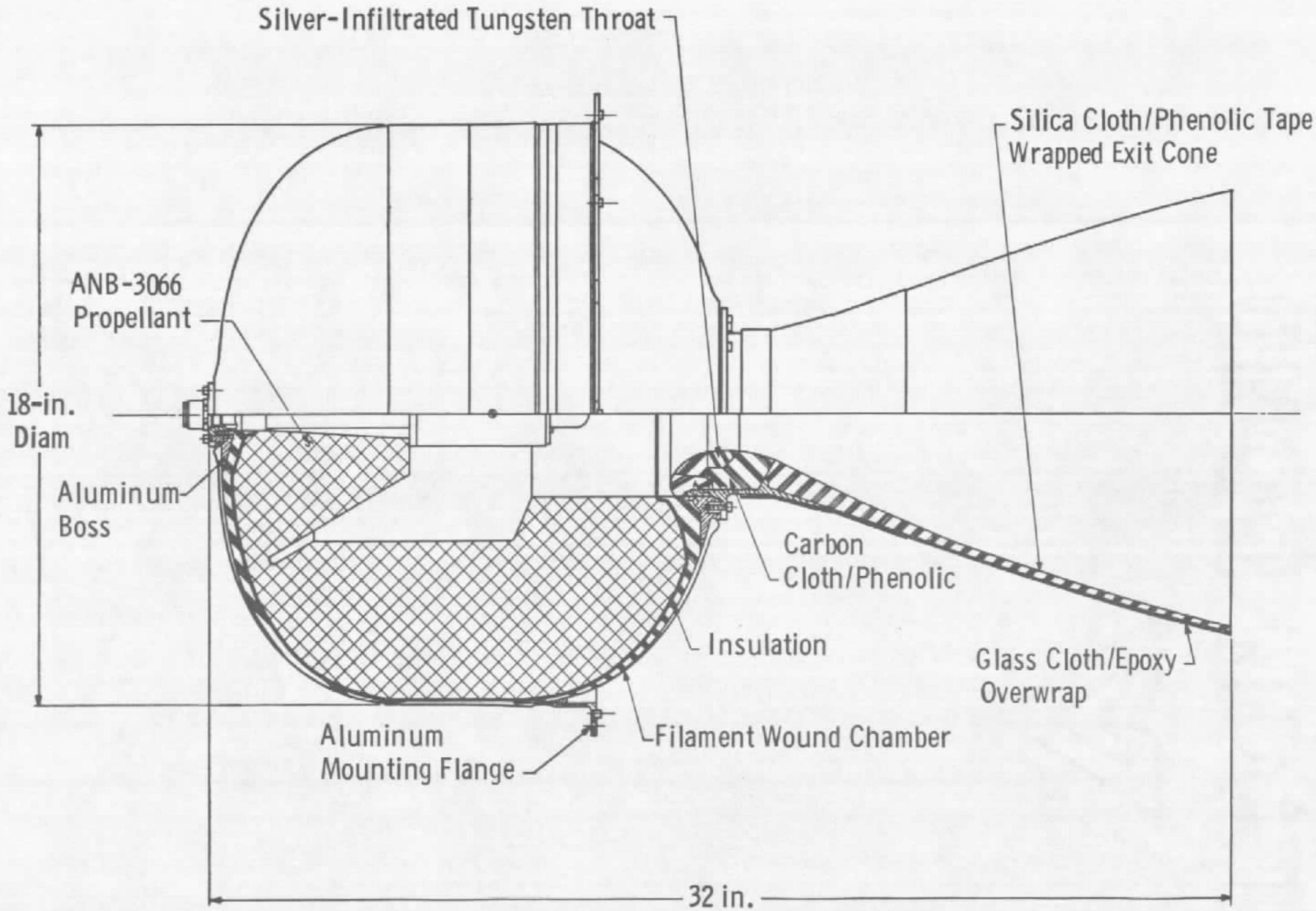
a. Artist's Sketch

Fig. 1 COMSAT INTELSAT II Communications Spacecraft



b. Photograph

Fig. 1 Concluded



a. Schematic

Fig. 2 AGC SVM-1 Solid-Propellant Rocket Motor

b. Photograph

Fig. 2 Concluded

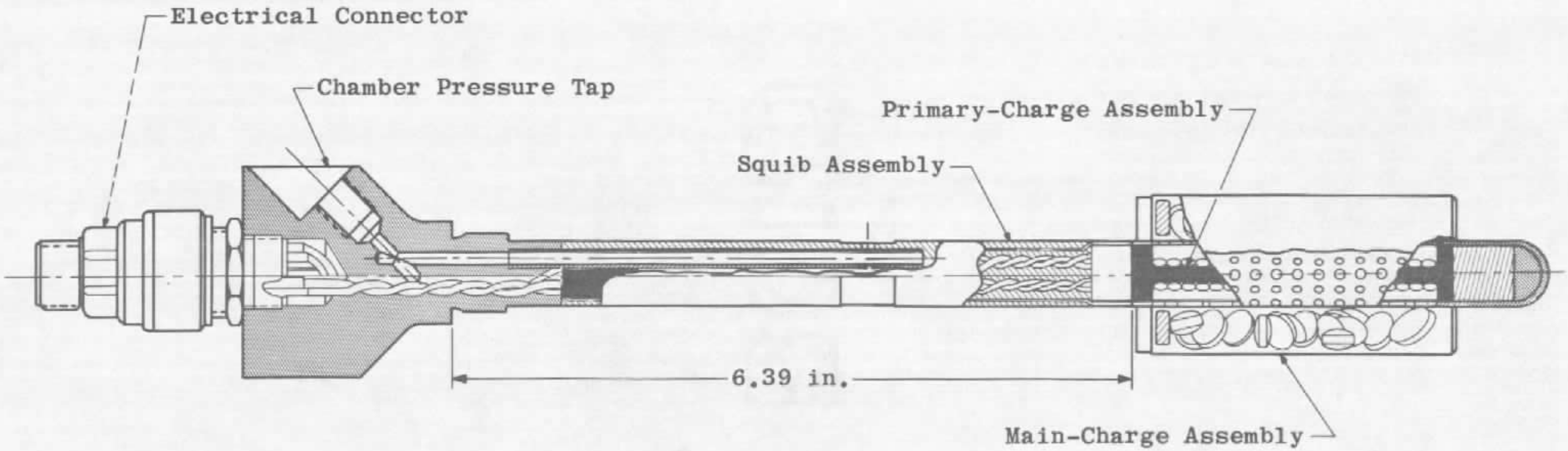
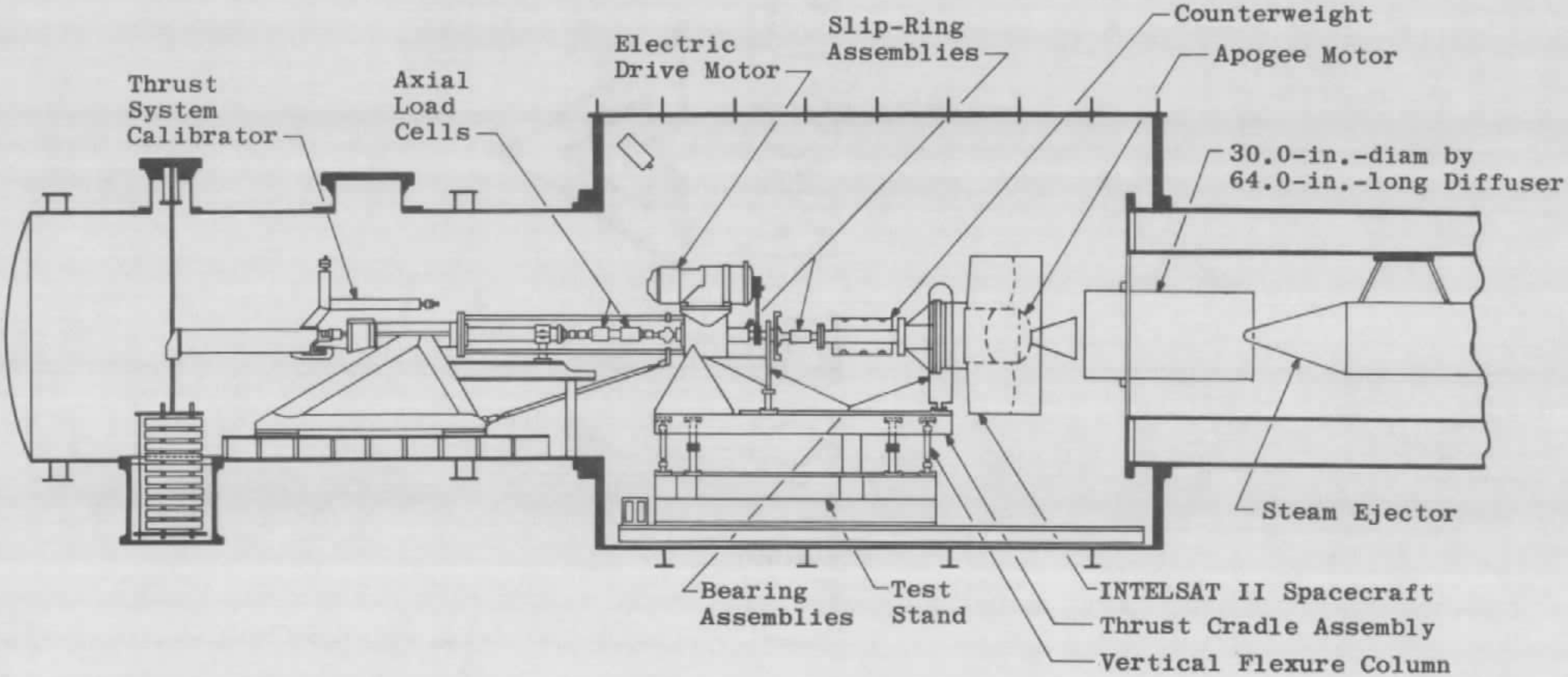
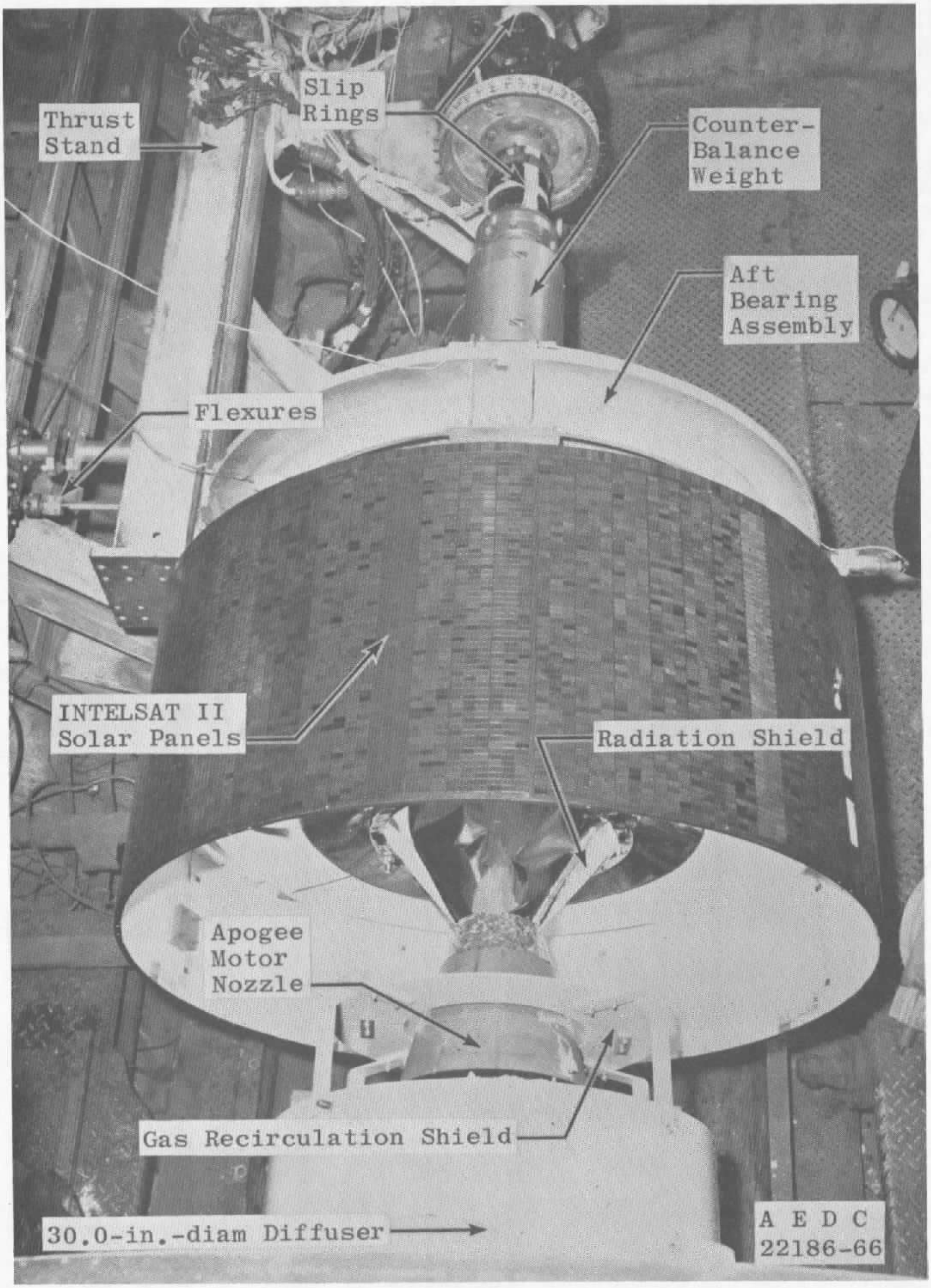


Fig. 3 Schematic of the AGC Igniter Assembly



a. Schematic

Fig. 4 Installation of the INTELSAT II Communications Spacecraft in the T-3 Test Cell



b. Photograph

Fig. 4 Concluded

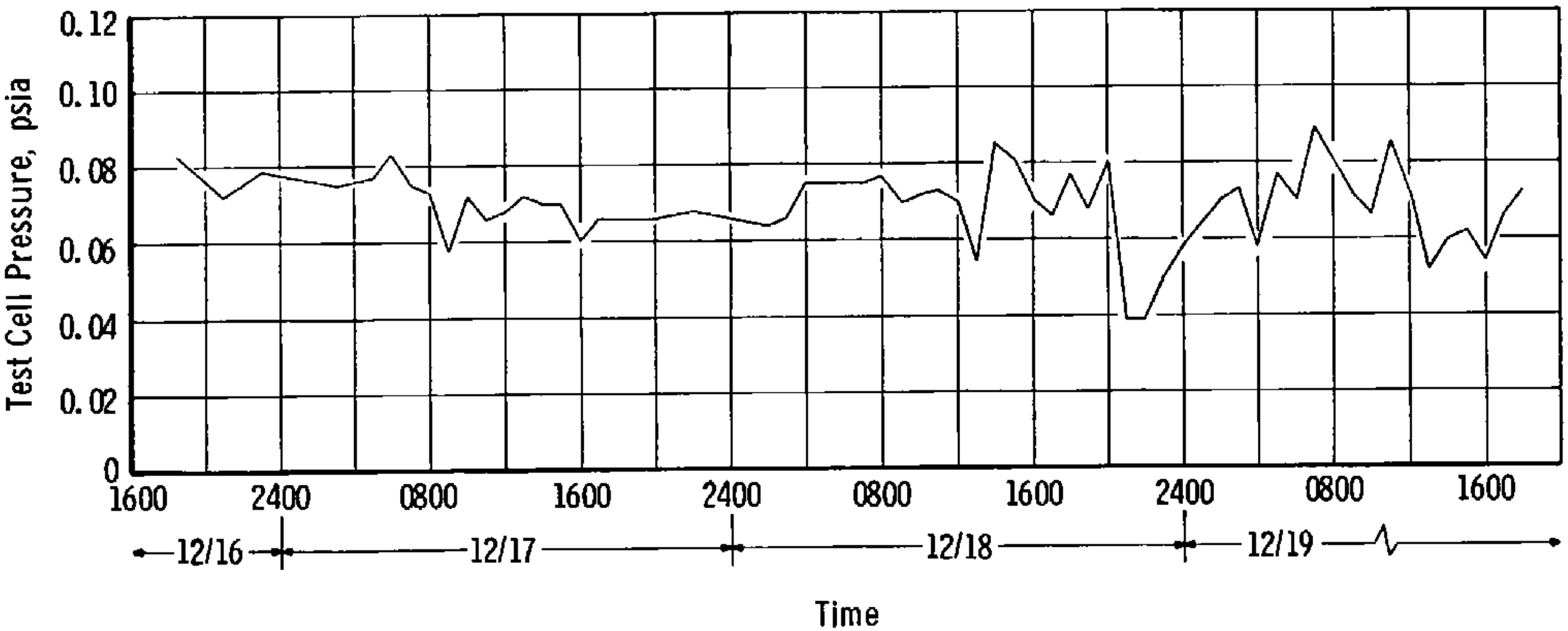
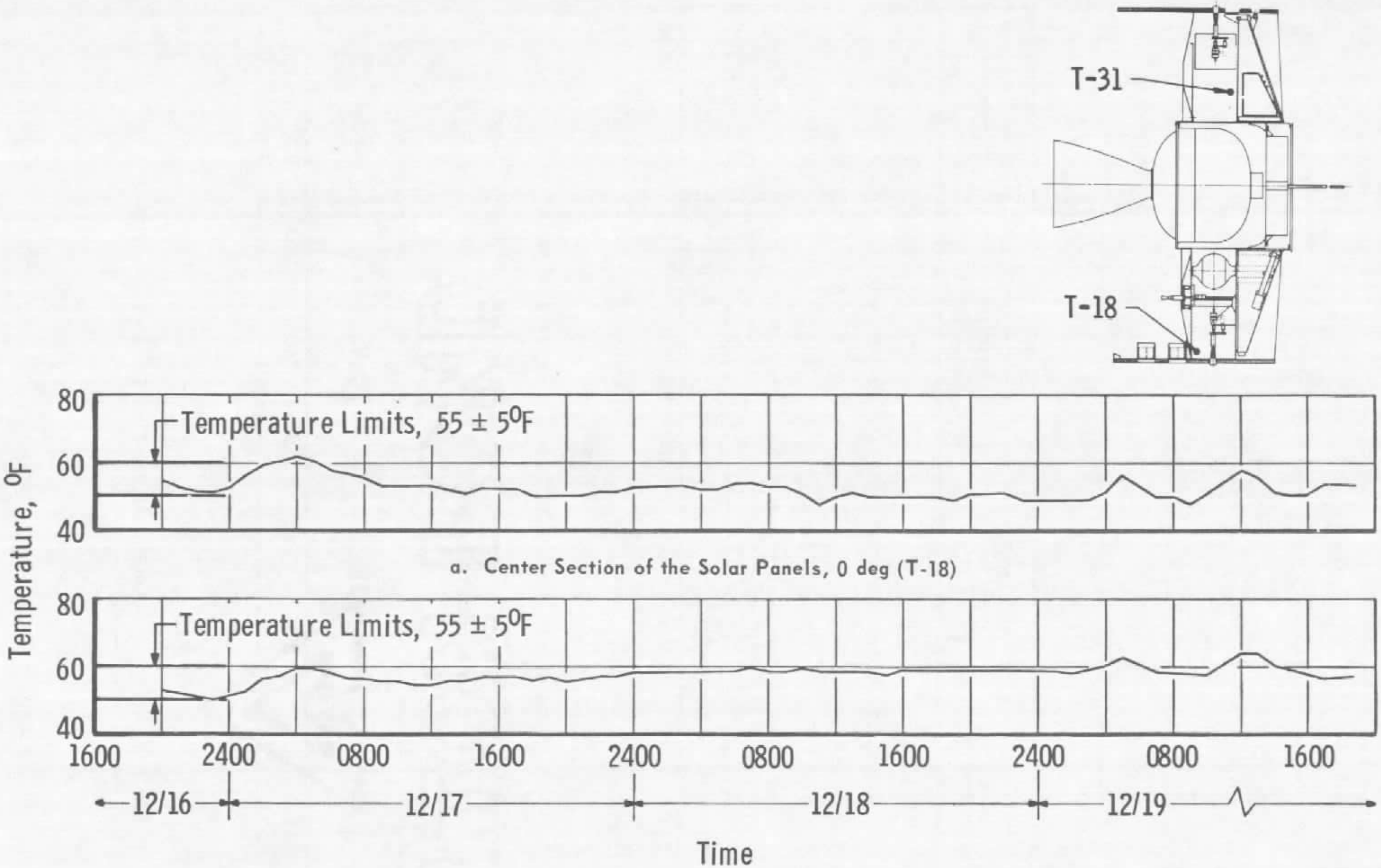


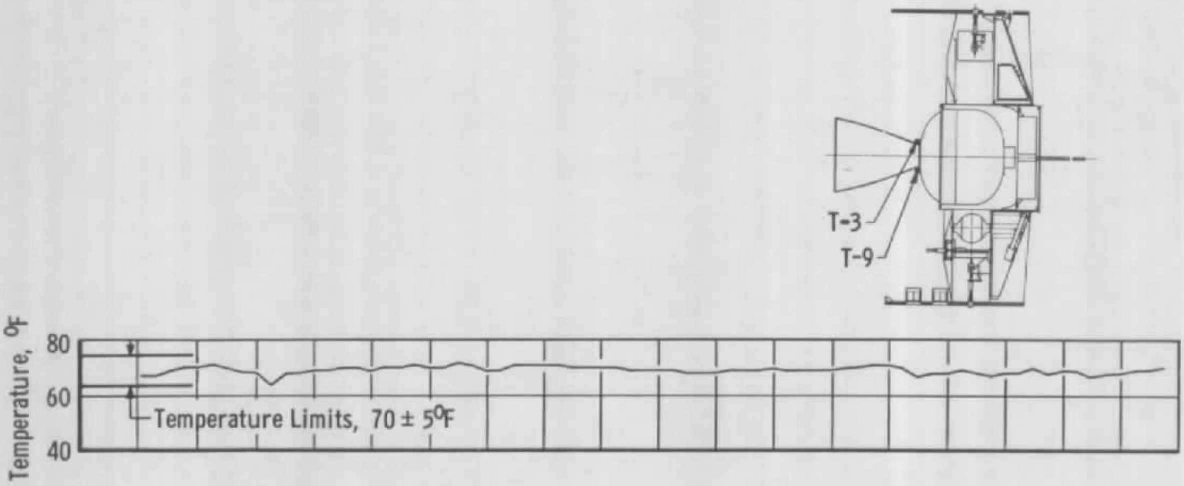
Fig. 5 Pre-Fire Test Cell Pressure-Time History during the Altitude Coast Period



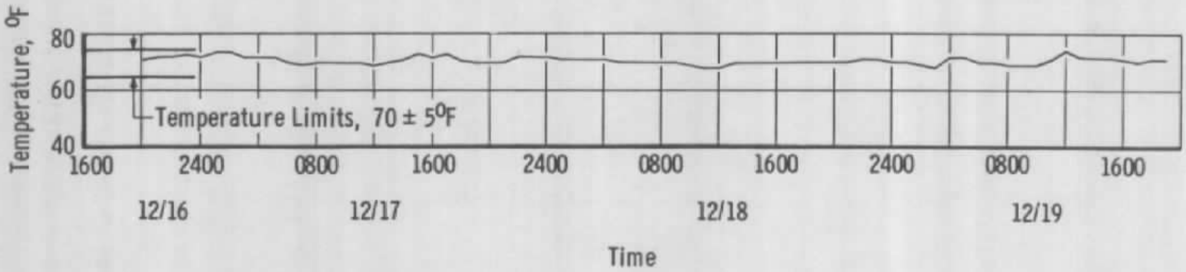
a. Center Section of the Solar Panels, 0 deg (T-18)

b. Temperature Sensor No. II, 185 deg (T-31)

Fig. 6 Pre-Fire Temperature-Time Histories of the INTELSAT II Spacecraft during the Altitude Coast Period



a. Nozzle Attachment Flange, 180 deg (T-3)



b. Nozzle Attachment Flange, 0 deg (T-9)

Fig. 7 Pre-Fire Temperature-Time Histories of the Nozzle Attachment Flange during the Altitude Coast Period

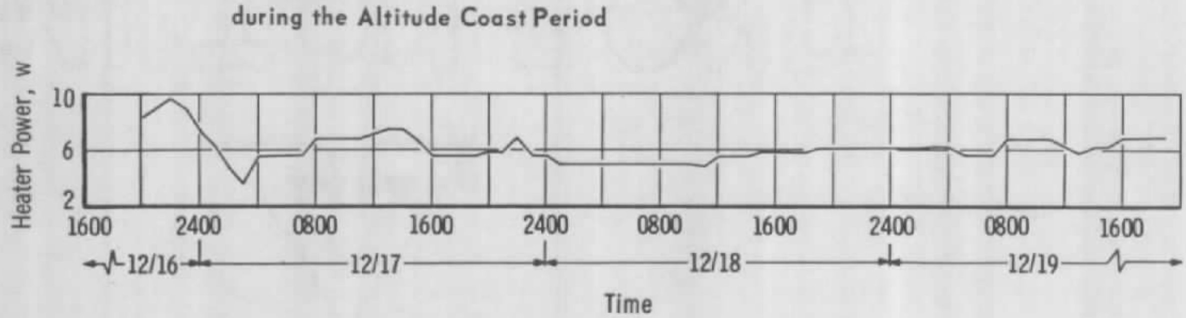


Fig. 8 Power Input to the Nozzle Attachment Flange Strip Heater

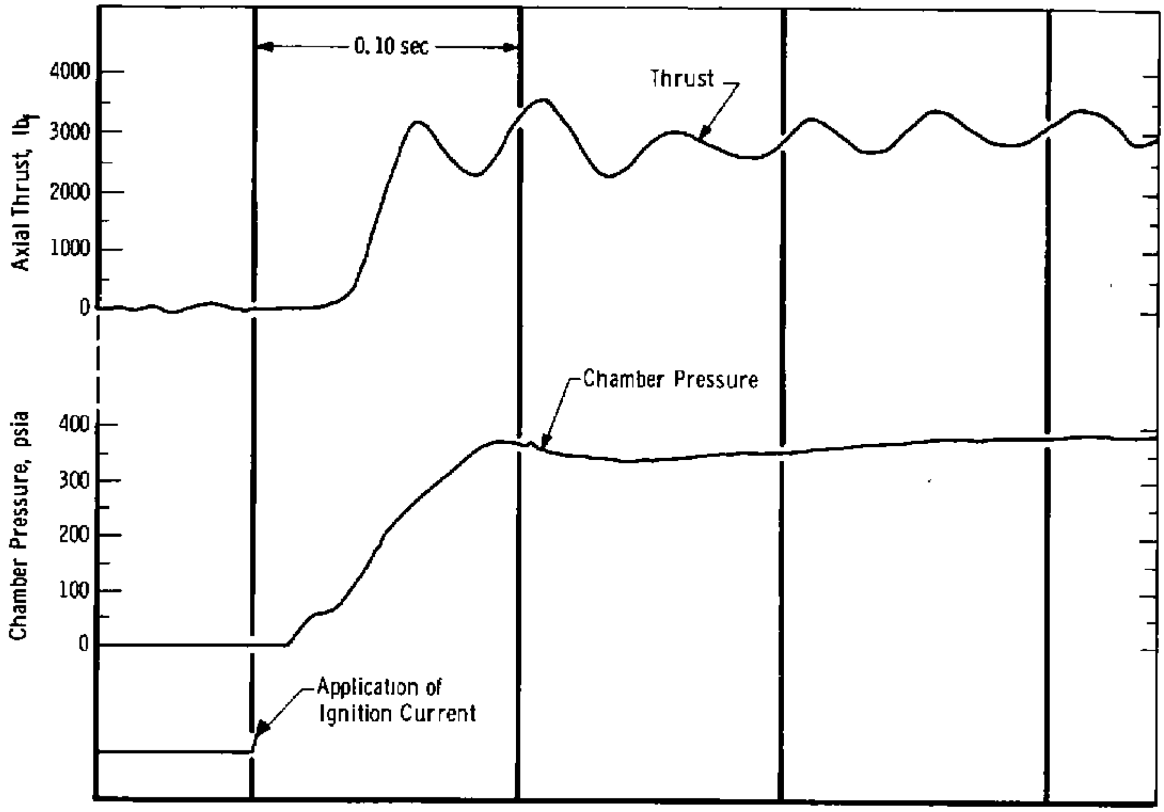


Fig. 9 Analog Trace of the Ignition Transient

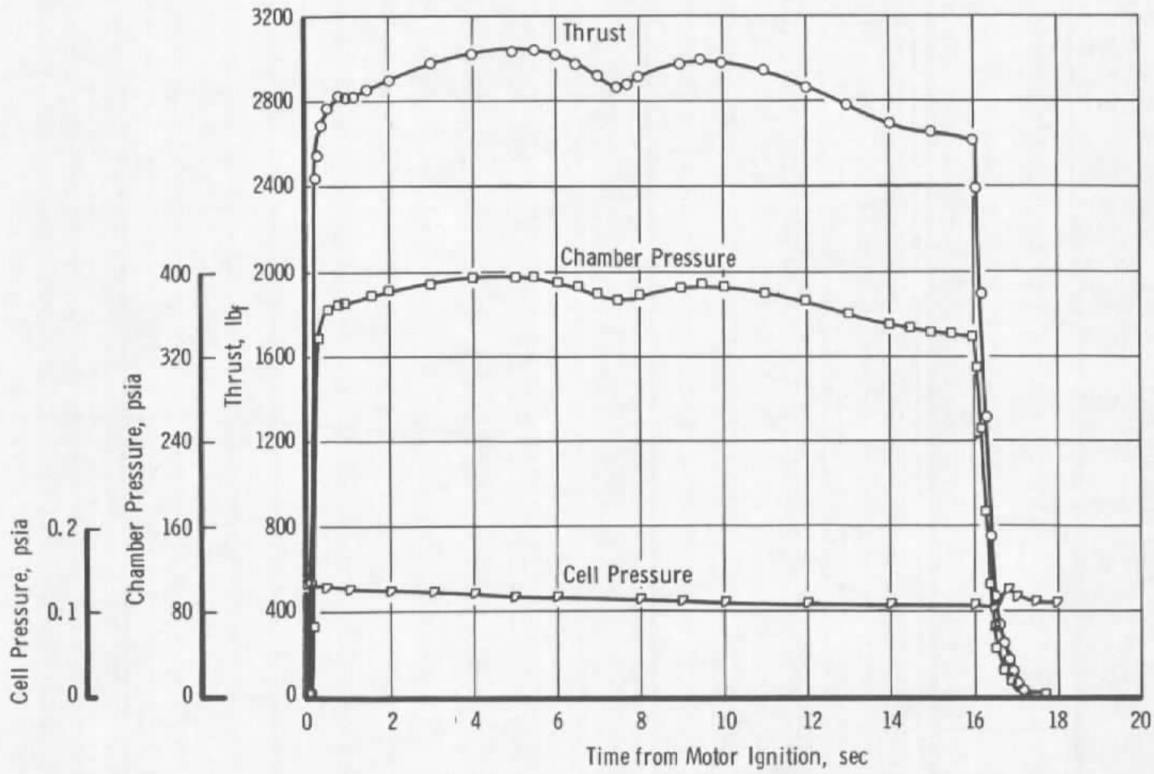


Fig. 10 Motor Performance Parameters versus Burn Time

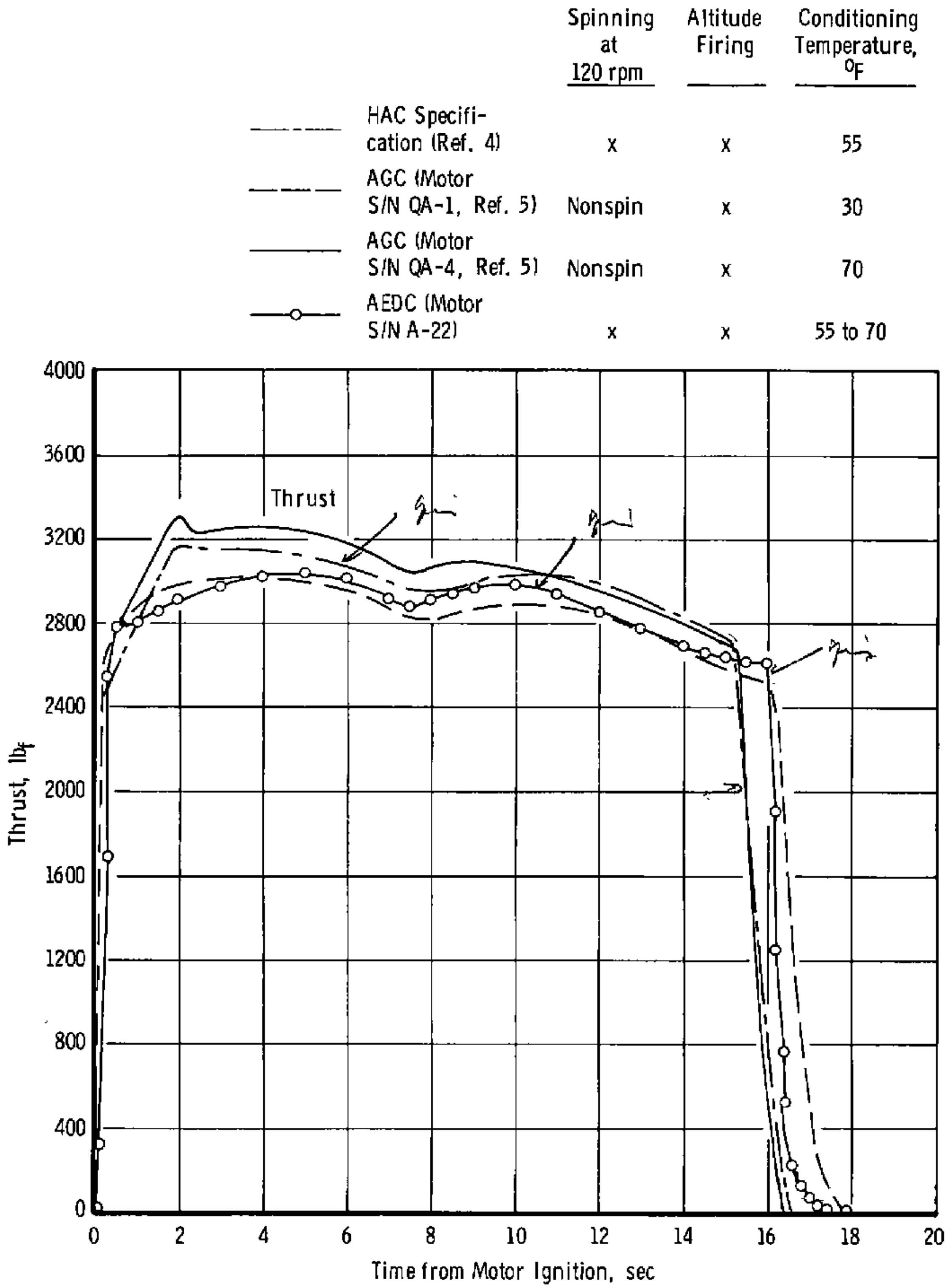


Fig. 11 Comparison of Motor Thrust with the HAC Specifications and Two AGC Firings

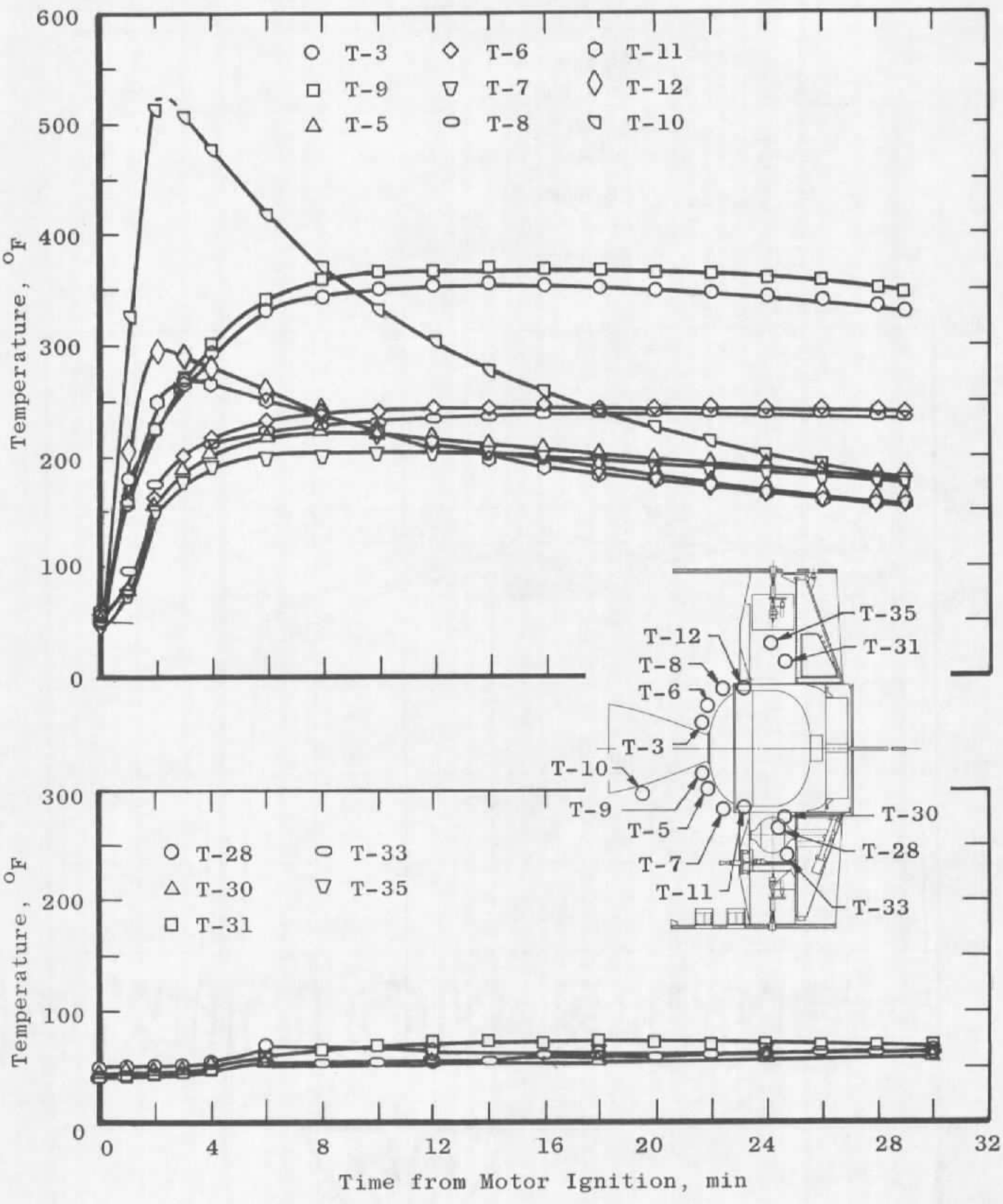


Fig. 12 Post-Fire Temperature-Time Histories of the Spacecraft and Apogee Motor

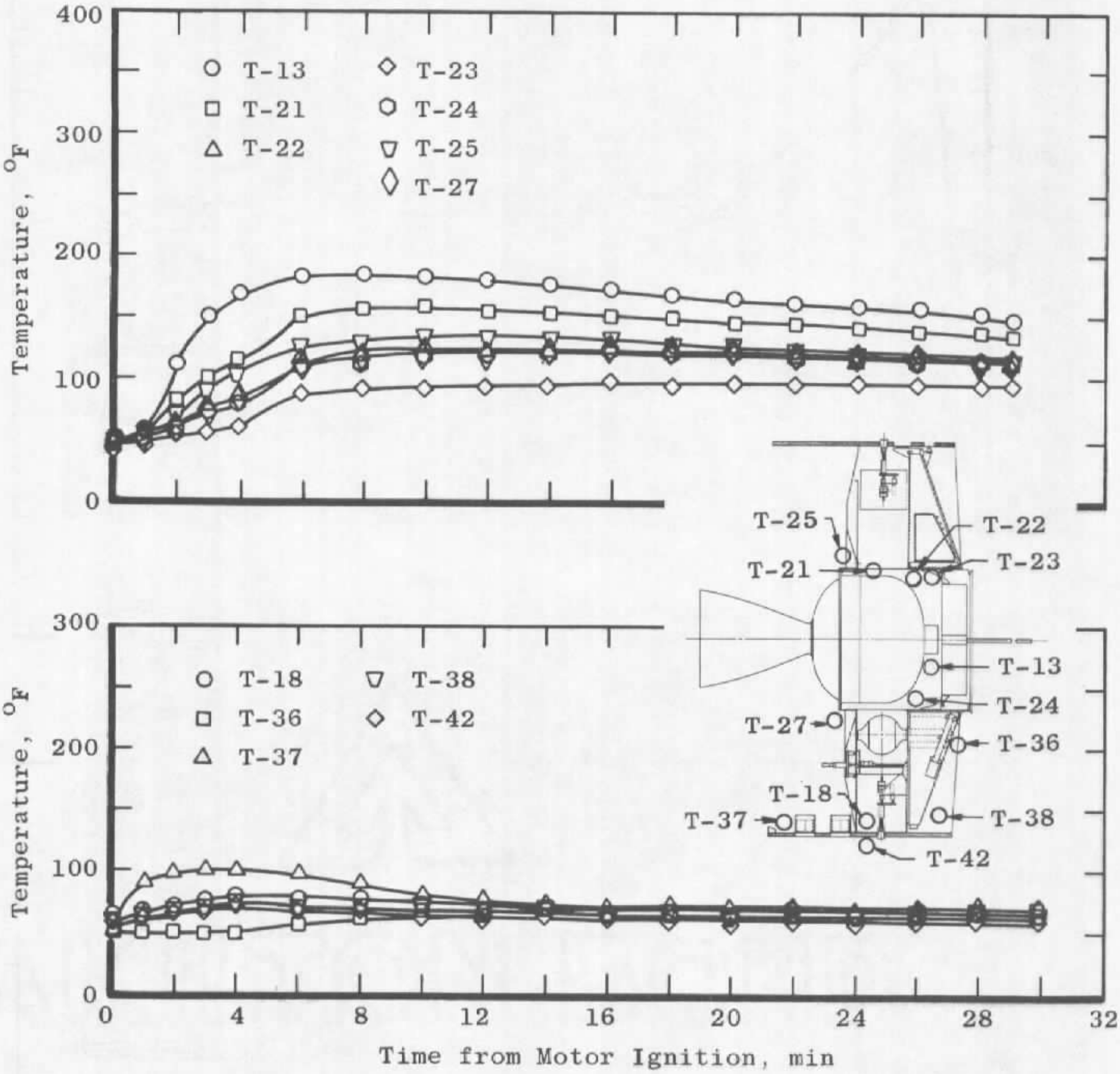
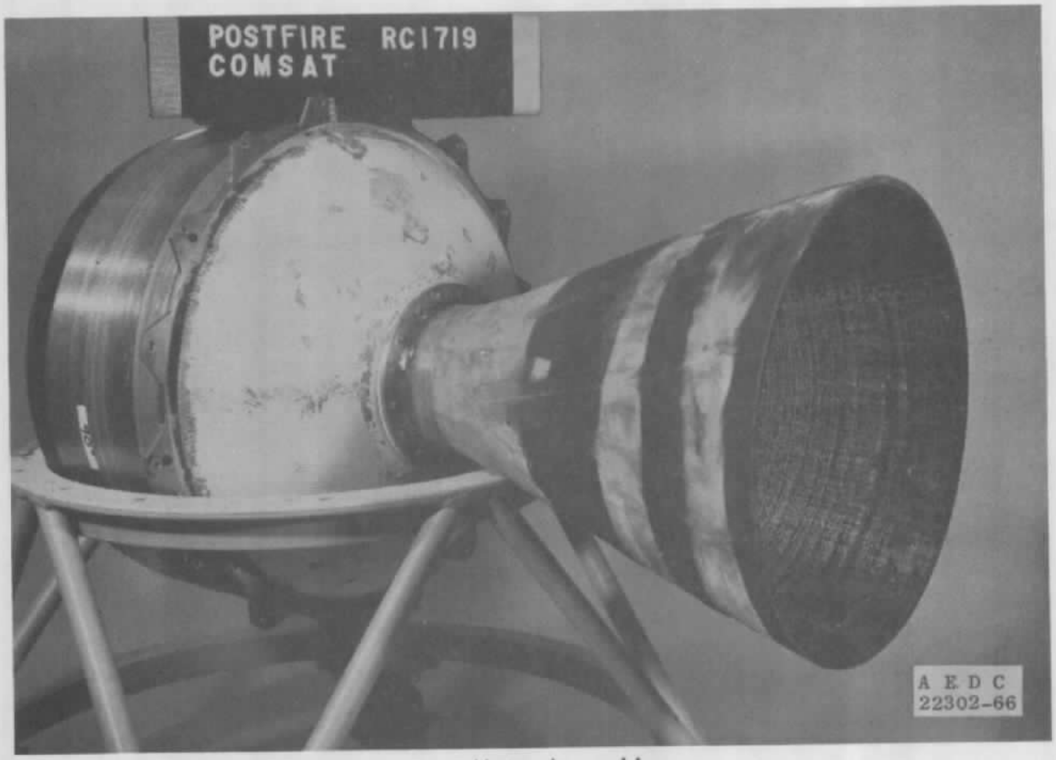
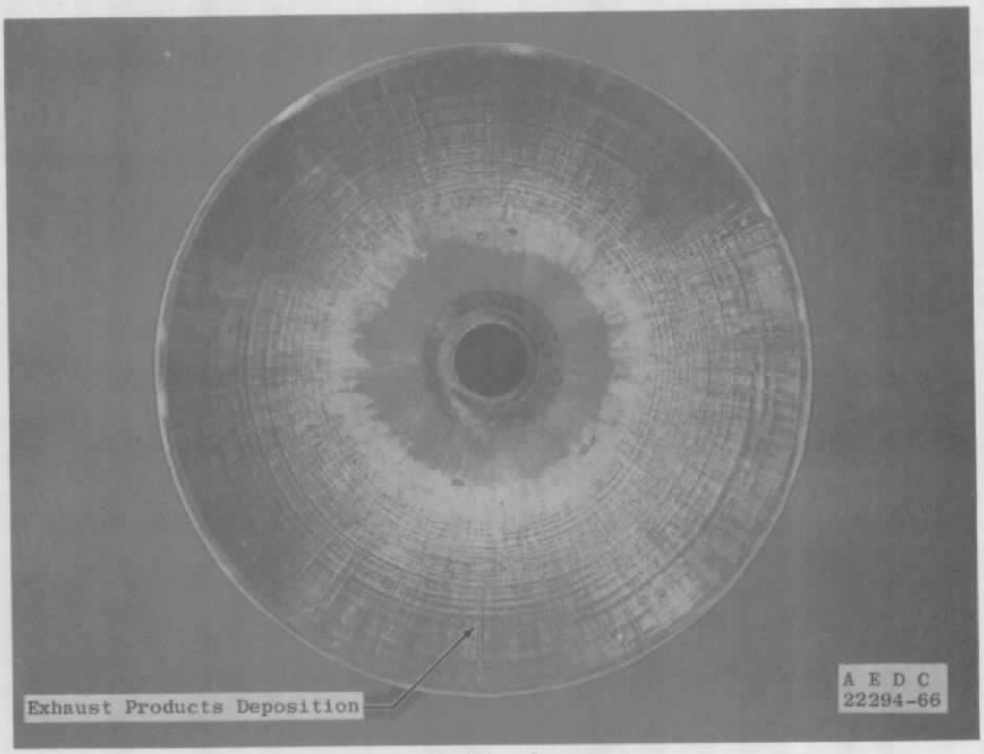


Fig. 12 Concluded

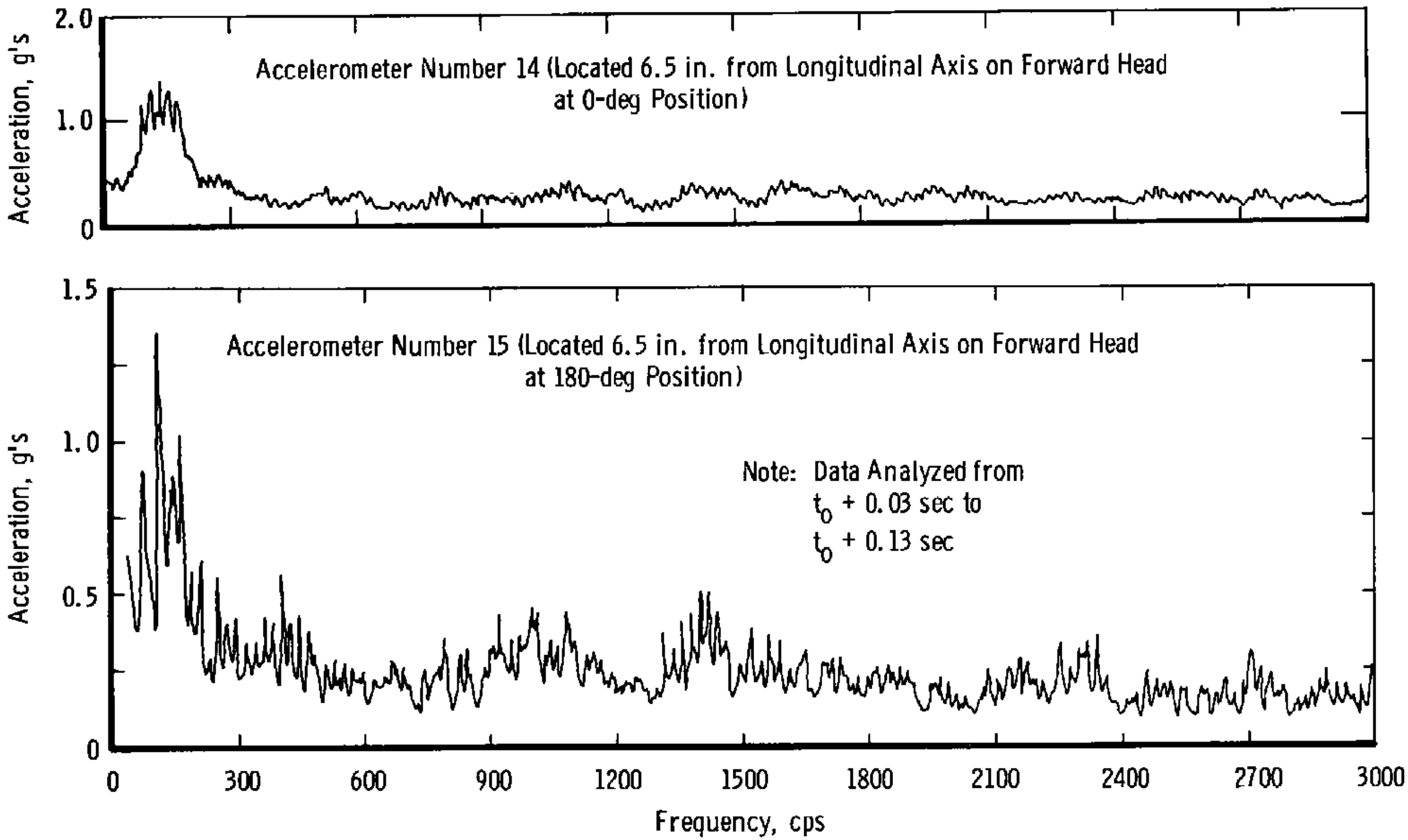


a. Motor Assembly



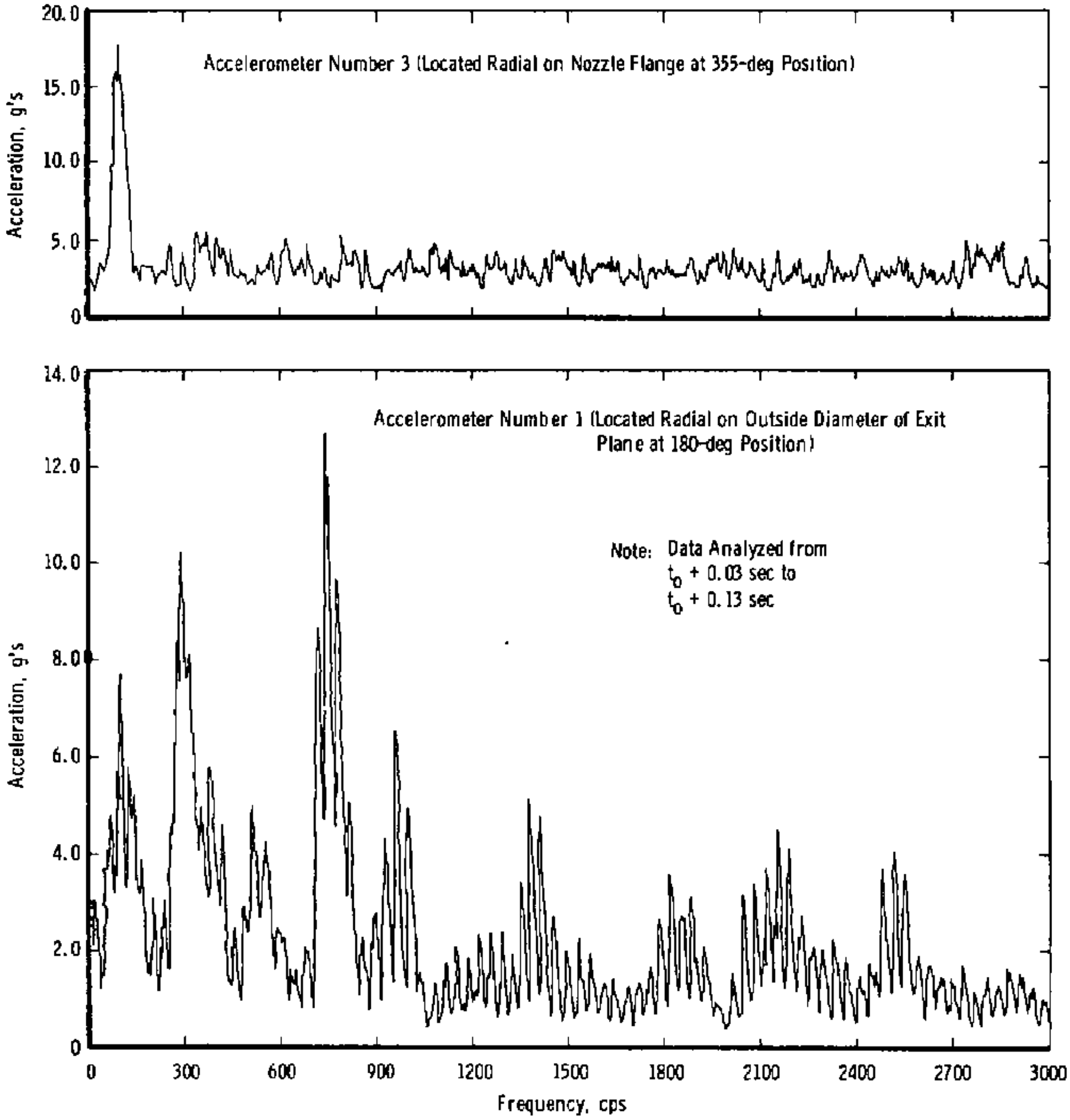
b. Nozzle

Fig. 13 Post-Fire Photographs of the Apogee Motor



a. Motor Case

Fig. 14 Vibrational Data at Ignition



b. Nozzle

Fig. 14 Concluded

TABLE I
SUMMARY OF MOTOR PHYSICAL DIMENSIONS

Motor Serial Number	A-22
Manufacturer's Stated Propellant Weight, lb_m	163.03
Expended Mass (AEDC) (Includes Igniter Weight), lb_m	165.507
Post-Fire Weight (Includes Motor Case, Nozzle, and Handling Ring), lb_m	35.760
Nozzle Throat Area, $in.^2$	
Pre-Fire	4.216
Post-Fire[1]	4.184
Percent Change from Pre-Fire	-0.77
Average	4.200
Nozzle Exit Area, $in.^2$	
Pre-Fire	139.98
Post-Fire[1]	139.53
Percent Change from Pre-Fire	-0.32
Average	139.76
Nozzle Area Ratio	
Pre-Fire	33.20
Post-Fire	33.35
Average	33.275

[1]Exhaust Product Deposition not Removed prior to Measurements

TABLE II
INSTRUMENTATION DESCRIPTION

Parameter	Estimated System Accuracy		Measuring Device	Range of Measuring Device	Recording Device	Method of System Calibration
	Steady-State at Operating Level	Integral, percent				
Axial Force, lb_f	±0.32 percent	---	Bonded Strain-Gage-Type Load Cells (2 used)	0 to 5000 lb_f	Millivolt-to-Frequency or Digital Converter onto Magnetic Tape	Deadweight
Total Impulse, lb_f-sec	---	±0.32				
Motor Chamber Pressure, psia	±0.84 percent	---	Bonded Strain-Gage-Type Transducers (2 used)	0 to 500 psia		Electrical
Chamber Pressure Integral, psia-sec	---	±0.83				
Test Cell Pressure, psia	±1.37 percent	---	Unbonded Strain-Gage-Type Transducers (3 used)	0 to 1.0 psia		
Test Cell Pressure Integral, psia-sec	---	±1.35				
Time Interval, msec	±2 msec	---	Synchronous Timing Line Generator	---	Photographically Recording Galvanometer-Type Oscillograph	Compare with 60 cps
Temperature, °F	±3°F	---	Copper-Constantan Thermocouples	0 to 500°F	Digital Millivoltmeter onto Magnetic Tape	Known Millivolt Source and NBS Temperature Tables
Weight, lb_m	±0.02 lb_m	---	Beam Balance Scales	0 to 400 lb_m	Visual Readout	Periodic Deadweight Calibration

TABLE III
THERMOCOUPLE AND ACCELEROMETER LOCATIONS

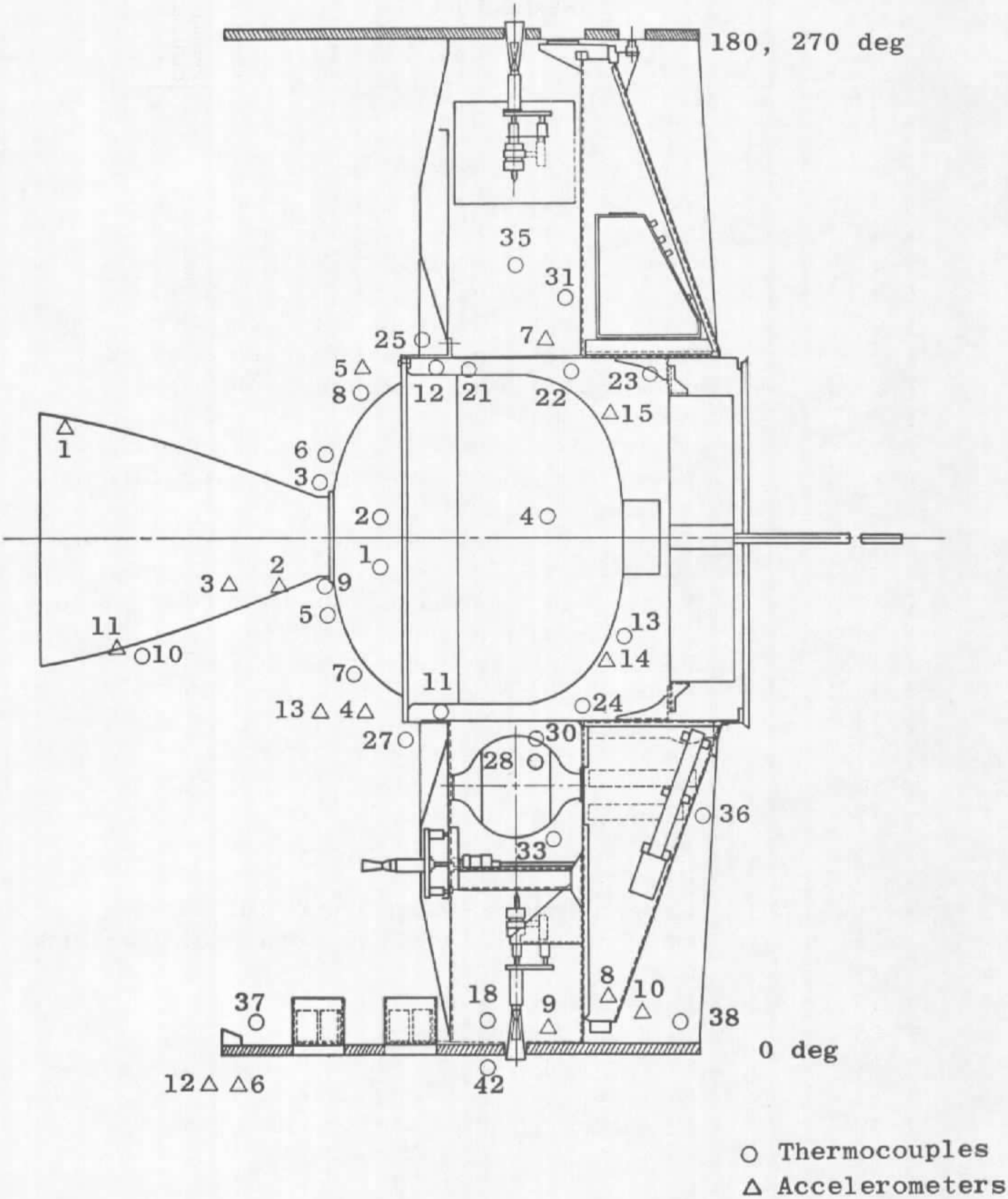


TABLE III (Continued)

THERMOCOUPLES

Thermocouple No.	Location, deg
T-1	Aft Face of grain bore, 0
T-2	Aft face of grain bore, 180
T-3	Nozzle flange, 180
T-4	Center face of grain plug, 180
T-5	Case wall (4.5 in. from bolt circle), 0
T-6	Case wall (4.5 in. from bolt circle), 180
T-7	Case wall (1.5 in. from bolt circle), 0
T-8	Case wall (1.5 in. from bolt circle), 180
T-9	Nozzle flange, 0
T-10	Outside exit cone (2.0 in. from exit plane), 0
T-11	Forward of attach flange (3.0 in. from forward attach plane), 0
T-12	Forward of attach flange (3.0 in. from forward attach plane), 180
T-13	Forward motor head (4.5 in. from bolt circle), 0
T-18	Center section of solar panels, 0
T-21	Inside thrust tube forward section, 270
T-22	Inside thrust tube center section, 270
T-23	Inside thrust tube aft section, 270
T-24	Inside thrust tube opposite T-22, 90
T-25	Inside motor flange, 180
T-27	Inside motor flange, 0
T-28	H_2O_2 tank (backup for T-30), 0
T-30	H_2O_2 tank, 0
T-31	By temperature sensor No. 2, 185
T-33	On aft bulkhead midway between solar panel and thrust tube, 5
T-35	H_2O_2 tank (opposite T-30), 180
T-36	Next to the traveling-wave tube, 0
T-37	On forward section of solar panels, 0
T-38	On aft section of solar panels, 0
T-42	On outside solar panel (opposite T-18), 0

TABLE III (Concluded)

ACCELEROMETERS

Accelerometer No.	Location, deg
A-1	Radial on outside diameter of exit plane (±141 g's), 180
A-2	Longitudinal on nozzle flange (±87 g's), 15
A-3	Radial on nozzle flange (±190 g's), 355
A-4	Longitudinal on end of thrust tube (±40 g's), 0
A-5	Longitudinal end of thrust tube (±40 g's), 180
A-6	Radial on solar panel area (±40 g's), 0
A-7	Longitudinal on midsection of thrust tube (±40 g's), 180
A-8	Radial on aft bulkhead (triax ±45 g's), 0
A-9	Longitudinal on aft bulkhead (triax ±35 g's), 0
A-10	Tangential on aft bulkhead (triax ±40 g's), 0
A-11	Tangential on exit plane (±40 g's), 0
A-12	Tangential on solar panel area (±40 g's), 0
A-13	Radial on end of thrust tube (±40 g's), 0
A-14	6.5 in. from longitudinal axis on forward head (±85 g's), 0
A-15	6.5 in. from longitudinal axis on forward head (±85 g's), 180

TABLE IV
SUMMARY OF MOTOR PERFORMANCE

Firing Number	1
Motor Serial Number	A22
Conditioning Temperature, °F	55
Test Date (1966)	December 19
Average Spin Rate, rpm	120.60
Ignition Altitude, ft	104,000
Ignition Lag Time (t_ℓ), sec	0.0150
Action Time (t_a), sec	16.60
Total Burn Time (t_t), sec	17.70
Measured Total Impulse (Average of Four Channels, Based on t_t), lb_f-sec	46,648
Maximum Deviation from Average, percent	0.01
Chamber Pressure Integral (Average of Two Channels, Based on t_t), psia-sec	6,077.4
Maximum Deviation from Average, percent	0.16
Cell Pressure Integral (Average of Four Channels, Based on t_t), psia-sec	2.0307
Maximum Deviation from Average, percent	0.83
Average Simulated Altitude (Based on t_t), ft	108,000
Vacuum Total Impulse (Based on t_t), lb_f-sec	46,931
Vacuum Specific Impulse, lb_f-sec/lb_m	
Based on Manufacturer's Stated Propellant Weight	287.87
Based on Expended Mass	283.56
Average Vacuum Thrust Coefficient (Based on t_t and Average Pre- and Post-Fire Throat Areas), C_f	1.839

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13 ABSTRACT

An Aerojet-General Corporation SVM-1 solid-propellant apogee rocket motor installed in an INTELSAT II communications spacecraft (S/N T-1) was fired, while spinning at 120 rpm, at a pressure altitude of 108,000 ft. The primary objective of the program was verification of vacuum performance of the apogee motor after a 75-hr exposure of the motor to a $55 \pm 5^{\circ}F$ temperature and a near-vacuum conditioning environment. Secondary objectives were to evaluate the control and performance of a strip heater attached to the nozzle flange, to check the structural integrity of a thermal shield on the motor aft dome, and to determine the existence of any significant dynamic coupling between the motor and the spacecraft. Motor performance is presented and compared with data from earlier firings of the same type of motor. Temperature-time histories at selected locations on the motor and spacecraft are also presented. Performance of the nozzle flange strip heaters, the structural integrity of the thermal shield, and vibration data are also discussed.

14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Intelsat II						
satellites, earth, artificial						
communications satellites						
environmental testing						
rocket motors						
apogee motors						
solid propellant						
temperature effects						
performance characteristics						